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DEPENDENCE OF SOLAR-THERMAL ROCKET PERFORMANCE ON CONCENTRATOR PERFORMANCE

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TRACT

aper introduces some performance parameters ant to rocket propulsion and shows how those ;o a rocket thruster powered by concentrated so- :rgy. Solar-Thermal Propulsion is a promising sion concept that could double spacecraft pay- apacity. Thrust and specific impulse (*Isp*), a ·e of the propellant usage efficiency, are defined paper. Elementary heat transfer theory is used w the upper limits in thrust and *Isp* that can ained. The configuration of a solar rocket that ired to perform an orbit transfer or other space ı is discussed. The basic geometry of off-axis loid concentrators is discussed along with a few on their concentration performance. An example for an 25 kw off-axis concentrator is discussed. ', a brief review of some more recent trade studies ce-deployed concentrators is included.

ꞮODUCTION

Thermal Propulsion is a promising propulsion t to move payloads from LEO (Low Earth Or- ıd up using relatively low thrust (Laug, 1989). ; done very efficiently by minimizing propellant required to perform an orbit transfer. It is pos- ıat a solar upper-stage thruster can deliver twice yload of a conventional upper-stage thruster.

use of sunlight as a propulsion energy source is ive because it allows a better choice of propel- This is because the propellant is not constrained . byproduct of a chemical reaction. In particular, hydrogen, which is very lightweight, may be used with no oxidizer, so that ground-launch weight is minimized. Thus, either more payload can be boosted or a smaller booster can be used resulting in cost savings for the Air Force, NASA, or anyone else who moves payloads from LEO into higher orbits.

The propellant must be heated to high temperatures to take advantage of its light weight. Thermodynamics allows temperatures approaching that of the Sun's surface (about 5800 K). This is well above the temperature achievable by practical large scale concentrators. It is also well above any material's melting or boiling point. Therefore, the propellant temperature is limited only by the performance of the concentrator system and/or the breakdown temperature of the materials making up the thruster.

The purpose of this paper is to give a feeling for the concentrator performance level needed to make this propulsion concept useful. This paper introduces some basic propulsion concepts. This basic prerequisite is necessary to show how concentrator behavior affects the performance of a solar rocket. Some concentrator basics are also included. Upper-limits are defined for performance of the solar rocket and compared with conventional rocket designs. The most popular configuration for a solar powered rocket is described and discussed. The geometry and performance of a specific type of off-axis paraboloid is discussed. Finally, recent concentrator structure development is reviewed briefly.

BASIC PROPULSION CONCEPTS

Rocket thrusters produce thrust by throwing mass, called the propellant, out a nozzle. Newton's second law gives the resulting thrust;

$$F = \frac{dp}{dt} = m\frac{dv}{dt} + v\frac{dm}{dt} \qquad 1$$

where p is the momentum of the propellant, v is the velocity of the propellant, and m is the mass of the propellant. Once the propellant has left the nozzle, and it is no longer accelerating, $\frac{dv}{dt}$ is zero. Therefore, the thrust is equal to the propellant mass flow rate out the nozzle times the final velocity of the propellant.

The two most important propulsion parameters are a rocket's thrust F and specific impulse Isp. Thrust is commonly referred to by the acceleration it will produce in terms of g, the acceleration due to gravity at Earth's surface. High thrust, greater than $1g$, is required to get a spacecraft off the Earth's surface. A $1g$ thrust force is equal to the total weight of the rocket, including payload and propellant. In the weightlessness of orbit very low thrust can be used.

The specific impulse Isp of a thruster is defined as the thrust divided by the mass flow rate out the nozzle and is a measure of how efficiently propellant mass is used. Isp is measured in meters per second (m/s) and in seconds (s) (the latter is actually thrust divided by weight flow rate).

It is desirable to maximize the Isp. The payload that can be carried by a thruster can be increased by increasing the Isp; or, less propellant is needed to boost the same payload if the thruster's Isp is increased. This latter advantage can allow a payload to be carried on a smaller booster from the Earth's surface, which will typically save many tens of millions of dollars. The former advantage can enable the delivery of payloads that could not otherwise be delivered on existing spacecraft.

The specific impulse is proportional to the square root of the propellant initial temperature T divided by the molecular mass m of the propellant.

$$Isp \propto \left(\frac{T}{m}\right)^{1/2} \qquad 2$$

This is a simplified but still the dominating relationship for Isp (Sutton, 1992). For molecular hydrogen, An Isp of 9800 m/s (1000 s) roughly corresponds with temperatures of around 3500 K, 8330 m/s (850 s) with temperatures of around 2500 K. Conventional rockets have Isp between 1960 and 4400 m/s, (200 and 450 s).

One other important propulsion system parameter is the total thruster mass. If the total mass of a solar thruster with concentrator is too great, then there is nothing to be gained by increasing Isp since the decrease in propellant mass is offset by increased thruster mass. Concentrator support structures must be lightweight and still stiff enough to maintain high solar concentration.

BASIC SOLAR CONCENTRATION CONCEPTS

We receive about 1350 watts per square meter (W/m^2) from the sun at our position in the vicinity of Earth. From our position, the mean angular diameter of the sun is 9.3 milliradians, or about 0.53 degrees. The irradiance, or power per unit area of the solar surface is about $63 \times 10^6\,W/m^2$. This is consistent with a surface temperature of about 5780 K. By the time sunlight reaches Earth, its intensity (W/m^2) is reduced by a factor of about 46,000, which is the ratio of the irradiance at the Sun's surface to the irradiance at Earth's orbit, $1350\,W/m^2$.

The function of a concentrator is to take solar radiation from the small patch of sky occupied by the sun and concentrate it into as small a spot, the focal point, as is possible. Ultimately, the maximum temperature achievable by a concentrator is the Sun's surface temperature, or 5780 K. This is essentially a statement of the second law of thermodynamics; for two objects in thermal contact, energy always flows from the hotter object to the cooler. An ideal concentrator gives perfect thermal connection between the solar surface and the focal point.

The ideal concentrator, by reaching solar temperatures, puts the focal point at the solar surface as far as thermodynamics is concerned. An observer on either the sun or the focal point will see sunlight over 2π steradians (one hemisphere). Therefore, an alternate and equivalent function of a concentrator is to fill the view from the focal point with as much sunlight as is possible.

The power emitted by a surface per unit area per unit steradian is defined as the radiance $\frac{dI}{d\Omega}$. The radiance of the sun is $63 \times 10^6/\pi$ or about 20,000,000 watts per square meter per steradian ($W/m^2 sr$). From the Earth the solid angle subtended by the Sun is $\pi \sin^2 \theta_{sun}$, where θ_{sun} is the half angle of the sun, or 9.3/2 milliradians. The power P received by an area A is;

$$P = \Omega A \frac{dI}{d\Omega} \qquad 3$$

where Ω is the solid angle of the radiating source as viewed by the area A. Therefore, a patch of area one meter square will absorb $20 \times 10^6 \times \pi \sin^2 \theta_{sun} \times 1m^2$ or about $1350\,W/m^2$ as stated earlier. On the surface of

n θ_{sun} is $\pi/2$ giving 63×10^6 watts on the same meter.

radiance $\frac{dI}{d\Omega}$ of the sun (or any other uniformly ing object) can never be increased by reflection action through an optical system (This is again lt of the second law of thermodynamics). There- ıe only way to increase absorbed power per unit t the focal point of a concentrator is to increase lid angle Ω of the sun as observed from that focal This is the same as filling the sky with concen- surface and the concentrator surface with sun- The practical result of this fact is that to achieve intensities or temperatures the concentrator sys- ıust fill more of the sky, as observed from the focal with sunlight. The other practical result of this a restatement of the second law of thermodynam- .e maximum intensity or temperature is achieved the hemisphere observed by the focal point is filled ncident sunlight (might as well be on the sun). A t concentrator effectively puts the focal point in .ive equilibrium with the Sun's surface.

: solid angle of a concentrator system can be cal- :d by:

$$I(\theta_{rim}) = \int_0^{2\pi} d\phi \int_0^{\theta_{rim}} d\theta \frac{dI}{d\Omega} \cos\theta \sin\theta \qquad 4$$

: $I(\theta_{rim})$ is the intensity at the focal point from centrator that has a rim half angle of θ_{rim}. $\frac{dI}{d\Omega}$: radiance or power per unit area per unit solid of the sun, 20×10^6 watts per meter squared per lian. Spherical coordinates are used and the con- ator rim is assumed to have rotational symmetry ; the z axis. Since the intensity of the solar disk is uniform, $\frac{dI}{d\Omega}$ is constant here:

$$I(\theta_{rim}) = 2\pi \int_0^{\theta_{rim}} d\theta \frac{dI}{d\Omega} \cos\theta \sin\theta \qquad 5$$

$$I(\theta_{rim}) = \frac{dI}{d\Omega} \pi \sin^2 \theta_{rim} \qquad 6$$

e limit where $\theta_{rim} \rightarrow \pi/2, \Omega \rightarrow \pi$ steradians in- of the 2π steradians in a hemisphere. This is ıse the surface receiving the radiation projects an $A\cos\theta$ and for large θ the power received per unit is reduced by $\cos\theta$ (as seen in Equation 4). The ;ive solid angle available for radiation on a flat sur- is only π steradians. The solid angle of the sun is $\pi \sin^2 \theta_{sun}$ as stated earlier. Equation 6 also works he Sun when $\theta_{rim} \rightarrow \theta_{sun} = 9.3/2$ milliradians as n previously.

ıe geometric concentration ratio is defined to be hadow area of the collecting optics divided by the area of the spot of sunlight focused at the focal point (A_c/A_f). For an ideal concentrator in the vicinity of Earth, this area is 46,000 times smaller than the shadow area of the collecting optics. The geometric concentration ratio is therefore 46,000 to 1, which is the maximum allowed by the second law of thermodynamics. Unfortunately, there is no such thing as an ideal concentrator.

The most common shape for a concentrator is a paraboloid. A paraboloid is the surface of rotation that results from rotating a parabola around the line between its focus and vertex. This line of symmetry is known as the optical axis. A paraboloid has the desirable property of producing a perfect point image, at the focal point, of a point on the optical axis an infinite distance away. The distance between the vertex and the focus, or focal point, is the focal length f. Extended objects are not perfectly focused, however. Astronomical telescopes use paraboloids to image extraterrestrial objects. These tend to be much longer focal length and produce much better images than systems used to collect solar energy. The short-focus, off-axis parabolic concentrators to be used for propulsion do not form good images.

Paraboloids produce at best an average geometric concentration ratio one fourth of the theoretical maximum 46,000 allowed by thermodynamics(Winston, 1991)(averaged over the focused spot area). Each mirror element of the paraboloid projects a cone of light of angular diameter $2\theta_{sun}$ at the focal point. This cone in turn projects an ellipse of light (a stretched image of the sun) on the focal plane. The integration over the concentrator surface of all projected ellipses onto the focal plane gives the intensity profile of sunlight on the focal point. Not all points on the paraboloid are at equal distance to the focal point so that the projected ellipses from different parts of the paraboloid do not overlap well. Therefore, the circle enclosing all the focused light has a larger area than would an ideal concentrator and so a paraboloid produces a lower geometric concentration ratio.

It is possible however, to achieve peak intensities that approach the thermodynamic limit of 46,000 suns. The intensity profile about the focal point of a paraboloid looks somewhat like a gaussian distribution. The geometric concentration ratio is an average intensity over this distribution divided by the intensity of 1 sun (1350 W/m^2). The focal point, where the center of the Sun is perfectly imaged, receives sunlight from all regions of the paraboloid. The intensity there is the solid angle of the concentrator multiplied by the solar radiance (see Equation 6). The concentration ratio at the

focal point is;

$$R_c = \frac{I(\theta_{rim})}{I(\theta_{sun})} = \frac{\sin^2\theta_{rim}}{\sin^2\theta_{sun}} = 46,000\sin^2\theta_{rim} \qquad 7$$

This is the peak concentration ratio as opposed to the geometric concentration ratio. For a concentrator that covers a 45 degree half-angle;

$$R_c = 46,000/2 = 23,000 \qquad 8$$

The effect of errors in the surface shape typically increases the focused spot size and decreases intensity. The standard unit of measure for mirror surface errors in the solar concentrator business is the mean slope error. The mean slope error is the mean deviation in angle of the mirror surface from the desired surface slope. The distribution of slope errors for a concentrator can be found by measuring the slope error for every point on the mirror surface and binning the frequency of errors. This slope error distribution typically looks like a Gaussian curve and can be convoluted with the focused intensity distribution of the perfect mirror to get the actual focused intensity distribution. However, in many cases it is usually good enough to sum the squares of the error with the square of the width of the perfect image to get an estimate;

$$\sigma_{system} = \left(\sigma^2_{perfect} + (2\sigma_{slope})^2\right)^{1/2} \qquad 9$$

where σ_{system} is the focal plane intensity distribution width, $\sigma_{perfect}$ is the width of the intensity distribution for a perfect paraboloid, and σ_{slope} is the width of the slope error distribution. The slope error is multiplied by two because reflected light deviates by twice the slope error. This process decreases both the geometric concentration ratio and the peak intensity. A one milliradian mean slope error is considered very good for a solar concentrator system and will not overly degrade performance. For comparison, consider that the width of the Sun is much larger.

The reflectance of real mirrors is usually somewhere between 90 and 98 percent. Polished or vapor deposited aluminum has a reflectance of about 92 percent. Polished silver surfaces reflect around 98 percent but are easily corroded. Protective coatings can decrease reflectance slightly but also greatly improve lifetime for terrestrial mirrors. Reflectance losses decrease the intensity of the distribution of light in the focal plane but do not change its shape.

SOLAR ROCKET PERFORMANCE UPPER-LIMITS

Figure 1 is a schematic drawing of a generic solar thruster. Sunlight enters from the left and is absorbed by the absorber/heat exchanger. Propellant flows in behind the window and proceeds to the right through the nozzle. Along the way, the propellant is heated in the absorber/heat exchanger. A window may be required for some designs to contain the propellant. The window spacer spaces the window back from the primary concentrator focal point at the front of the secondary concentrator. This minimizes the thermal flux in the window. The window spacer also serves as a heat shield to external components and can provide initial heating of the propellant. A Secondary concentrator may be used to increase the solar flux through the absorber. A cavity can be used to increase the absorptivity of the absorber/heat exchanger by making it look more like a blackbody.

A propellant temperature T can only be reached if the radiation field temperature T_{rad} exceeds T. T_{rad} is given by the Stefan-Boltzmann law;

$$T^4_{rad} = \frac{R_c I_{earth}}{\sigma} \qquad 10$$

where I_{earth} is the $1350\,\mathrm{W/m^2}$ received in Earth's orbit and R_c is the concentration ratio. If the propellant temperature T approaches T_{rad}, then combining the relationship for specific impulse (Equation 2) with Equation 10 gives;

$$Isp \propto R_c^{1/8} \qquad 11$$

This is the maximum possible specific impulse for a given concentration ratio R_c and is proportional to the eighth root of concentration ratio.

The temperature reached by a particular concentration ratio R_c can also be found from;

$$\frac{T^4}{T^4_{sun}} = \frac{RI_{earth}}{46,000I_{earth}} \qquad 12$$

$$T = T_{sun}\left(\frac{R}{46,000}\right)^{1/4} \qquad 13$$

where T_{sun} is the solar surface temperature. A concentration ratio of 10,000 will produce an equilibrium temperature (allowing only radiation transfer) of 3960 K. This is well above the 3500 K or so temperature needed to achieve $9800\,\mathrm{m/s}(1000\,\mathrm{s})$ Isp. If the concentration ratio is cut in half, then the Isp goes as the eighth power of 1/2; therefore, R=5000 gives 3600 K. This is sufficient to achieve $9800\,\mathrm{m/s}(1000\,\mathrm{s})$Isp. However, the thruster should not be operated near radiative equilibrium since this means a large amount of incident radiation is reradiated and lost. Operating at 5000 suns to get 2500 K on the other hand, results in only 22 percent reradiation loss. It is important to remember that

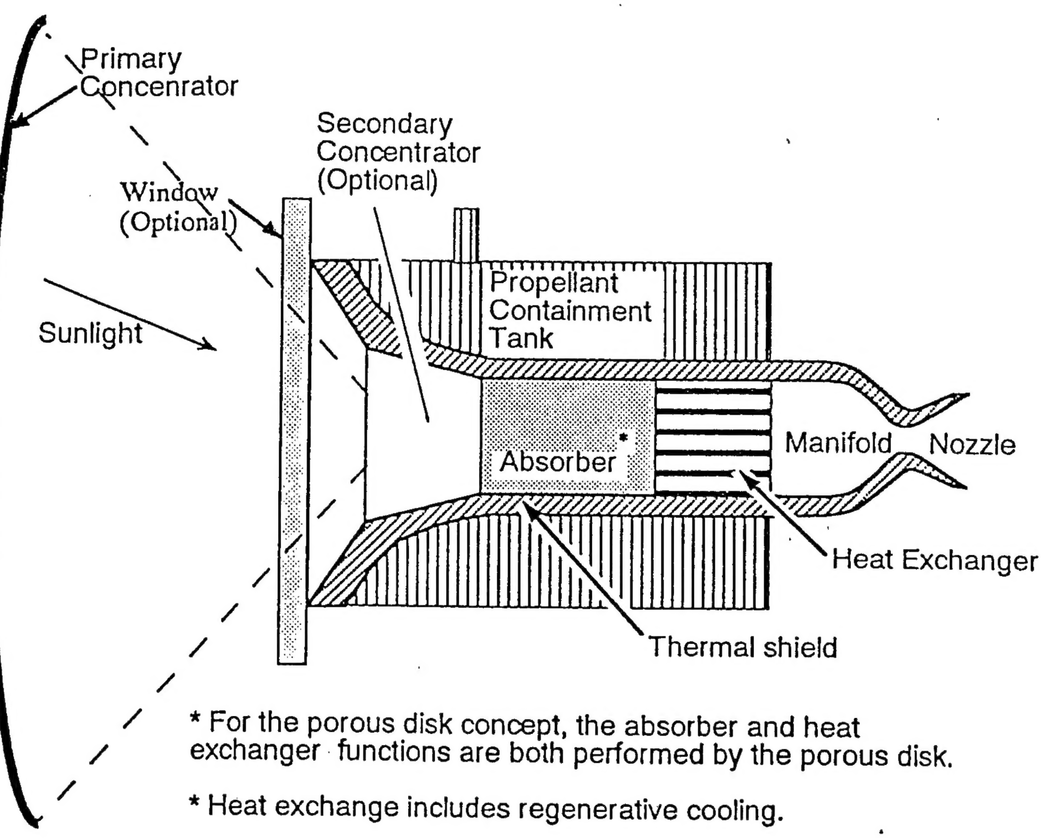


Figure 1: GENERIC SOLAR THRUSTER.

ese numbers for Isp are approximate and only meant a guide. Real propulsion nozzles may do better or ırse in either Isp or energy efficiency.

The upper-limit of propellant temperature also dends upon the details of the thruster's receiver design. cavity absorber absorbs all light entering its aperture, ereby losing information on the intensity distribution. e maximum temperature possible is that correspond- to the average intensity I_{avg};

$$I_{avg} = \frac{1}{A} \int \int_{Aperture} I(x,y)dxdy \qquad 14$$

ere A is the aperture area. This average intensity less than the peak intensity available. A different uster design could use the peak of the intensity disoution to do the final propellant heating. This type thruster could produce higher temperatures than a ity absorber. Another alternative is to use a secdary concentrator to fold the light in the wings of intensity distribution into the aperture of a cavreceiver. This approach squeezes all the light into maller aperture to increase the total intensity to a value possibly larger than the peak intensity of the original distribution. An absorber should be well matched to the distribution of focused light (i.e. a cylindrical cavity may not work well).

Thrust is given by Newton's Law, $F = v\frac{dm}{dt}$ (Equation 1), where v is the propellant velocity and $\frac{dm}{dt}$ is the propellant mass flow rate. The power P put into propellant flow is $\frac{1}{2}v^2\frac{dm}{dt}$. v^2 is proportional to the kinetic energy imparted to the propellant. If the Isp is held fixed, then v is also constant;

$$Isp \equiv F \Big/ \frac{dm}{dt} = v \qquad 15$$

(v is also proportional to the square root of T) and thrust F is proportional to $\frac{dm}{dt}$ only. $\frac{dm}{dt}$ is proportional to the thermal energy input since that determines how much propellant per unit time can be heated to the required temperature. Therefore, thrust is proportional to the area of the concentrator (for fixed Isp).

If mass flow $\frac{dm}{dt}$ is held constant, then both thrust F and Isp will go up with increasing temperature. If

$\frac{dm}{dt}$ is increased while keeping the thermal power input constant, thrust increases monotonically but the propellant temperature will drop. Simultaneously, receiver temperatures should drop, reducing reradiation and increasing thermal efficiency. Also, there are energy loss mechanisms involved with the energy states of molecular hydrogen that increase with increasing temperature. Thrust can be traded off with *Isp*. Figure 2 shows approximate thrust and *Isp* as a function of temperature. Higher *Isp* decreases thrust very rapidly. However, higher temperatures are always desirable if the mass flow rate can be maintained.

Figure 3 shows performance envelopes for a variety of propulsion systems. The specific impulse is plotted on the vertical axis and the thrust to vehicle weight ratio is plotted on the horizontal axis. Conventional rocket thrusters are in the oval labelled "Liquid and Solid Chemical Propellants". These typically produce thrust between 0.1 to a few hundred *g*s. The maximum *Isp* is about 4500 m/s. The region of this chart where boosting from the Earth's surface is possible is on the right side of 1 g. As is well known, conventional thrusters can leave the ground. Solar-Thermal propulsion however cannot as its oval lies to the left of 0.1 g. The top of the Arcjet, Solar-Thermal, Nuclear Fission, and Laser-Thermal propulsion ovals is limited to around 10,000 m/s because of the limitation in fabrication materials; no material can survive the temperatures required to reach higher *Isp*. Electrostatic and electromagnetic thrusters do not have this constraint because they use electromagnetic fields to contain the hot propellant (in this case a plasma).

SOLAR ROCKET CONFIGURATION

Operation of a Solar Rocket requires pointing the thruster force vector along the desired direction of acceleration and pointing the concentrator system at the sun at the same time. The thrust vector passes through the center-of-mass of the rocket (unless you want only to go in loops) and preferably down the centerline of the rocket body (so that propellant usage does not offset the center of mass). The rocket may be rotated around the thrust vector, which should be the body axis, without changing the direction of acceleration. Only one other degree of freedom is then required to allow the concentrator system to collect sunlight at all times. This is obtained by allowing the concentrator system to pivot on an axis perpendicular to the body axis.

The nozzle produces thrust by flowing hot propellant, such as molecular hydrogen, through its throat. The path of the hot propellant to the nozzle throat must be kept short to minimize thermal conduction losses and

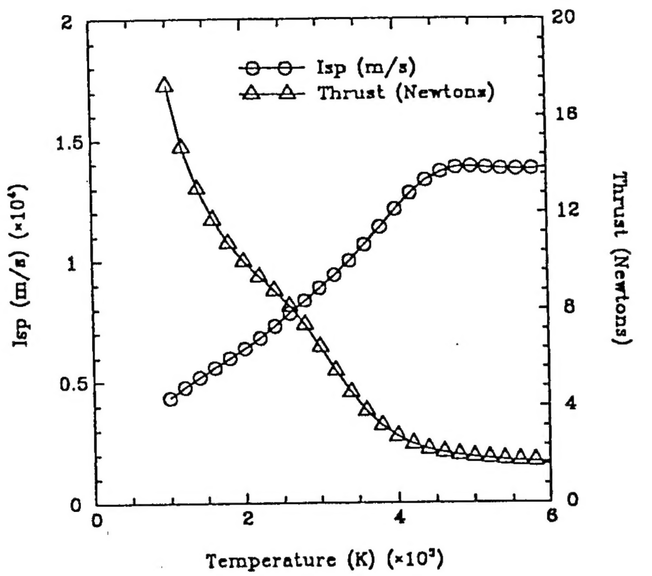
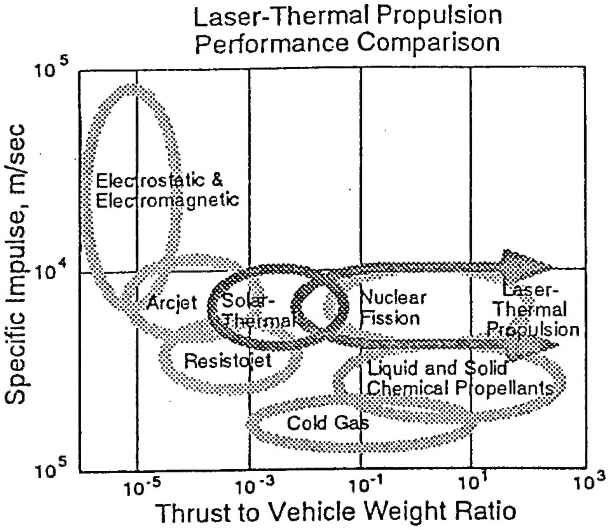


Figure 2: TRADEOFF OF THRUST AND *Isp* WITH CHANGING PROPELLANT TEMPERATURE.

spacecraft weight. The absorber/heat exchanger must therefore be very close to the nozzle. Furthermore, the exhaust plume, which expands out at very high angles from the nozzle, can do serious damage to concentrator structures (Merkle and Yu, 1988). The concentrator must be removed from this hazard.

The only way to meet the above constraints is to use off-axis concentrators. Sunlight enters the concentrator system and is deflected at a large angle into the absorber/heat exchanger. The direction to the sun is somewhere close to 90 degrees from the optical axis of the concentrators. A paraboloid shape is still desirable since it focuses an on-axis point into the focal point. The focused light from the Sun comes into the thruster bound by the surface of a cone. The surface of this cone is far removed from the plume to prevent plume impingement damage to the concentrator. The centerline of this cone, which is the optical axis, is close to perpendicular to the rocket body axis to allow the concentrators to be rotated to give the second degree of freedom required. The resulting appearance of most solar powered thruster designs is similar to that of a conventional rocket but with two large opposing concentrators attached to a point near the rocket nozzle (See Figure 4).

There are other ways to build a solar rocket if one is willing to reduce some of the above constraints. One is to use a focal point far from the concentrator and transmit the solar energy to the thruster via a fiber optic



'igure 3: PERFORMANCE ENVELOPES FOR A VA-
:IETY OF PROPULSION SYSTEMS.

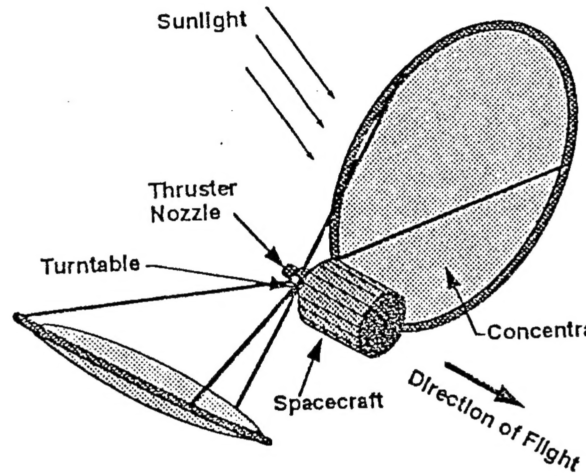


Figure 4: SOLAR ROCKET CONFIGURATION.

able (Nakamura and Irvin 1992). Another is to thrust nly in certain orientations, thus relaxing the need for wo degrees of freedom in concentrator orientation.

Conventional thrusters typically burn all of their proellant in a few minutes. It takes just this one short, ut high acceleration boost to place the spacecraft into transfer orbit. Once the spacecraft rises to its highst point, another burst of thrust puts the spacecraft ıto a circular orbit (if that is the desired orbit). A ɔlar-thermal thruster, on the other hand, does not have nough thrust to do a large orbit transfer with one short hrust. It will take many hours typically for a solar hruster to expell all of its propellant.

This adds further constraints on how a solar-thermal hruster must operate. A continuous but low accelertion thrust can be used to spiral the spacecraft to its rbit. This is not the most efficient way to perform a rbit transfer, however. A more efficient approach is ɔ thrust only at apogee or perigee. At these points ll energy is put into maximizing the increase in linear ıomentum and angular momentum. This type of orbit ıaneuver is called a multi-burn orbit transfer. In LEO Low Earth Orbit) these thrust periods will last around 0 minutes. Then the spacecraft must not thrust until he next perigee or apogee. This slows the orbit transer since the majority of time is spent waiting until the ext apogee or perigee. Most orbit transfers using solarhermal propulsion will require around two weeks.

THE OFF-AXIS PARABOLOID

Concentrators for most solar-thermal spacecraft concepts are extreme off-axis paraboloids as compared with typical terrestrial concentrators. While typically still projecting a circular cone of concentrated light, the intensity distribution within that cone is very nonuniform. The geometric concentration ratio is lower than available from an on-axis paraboloid concentrator because of the larger variation in distances from the reflective surface to the focal point. Finally, the shape of the concentrator itself (one that projects light within a circular cone) is elliptically shaped instead of circular.

The intersection of a circular cone and paraboloid, with the cone apex at the focal point of the paraboloid, is an ellipse that lies within a plane. This is convenient as it is usually simpler to build support structures that lie in a plane; for this reason a torus is the simplest support structure for inflatable and other types of solar propulsion concentrators that require support. Rigid self-supporting concentrators need not be constrained to this geometry.

Figure 5 shows the geometry and dimensions of a generic off-axis concentrator. The Y-Z plane is viewed in this figure and the X axis is perpendicular to the page and pointing at the reader. The Z axis is the axis of rotation symmetry of the paraboloid. The circular cone has a half angle of θ_c and its apex is at the focal point. The axis of symmetry of the cone is rotated in the Y-Z plane at an angle ϕ measured clockwise from the Z axis. The projection of the ellipse, or torus, along the axis of symmetry of the paraboloid is a circle. The

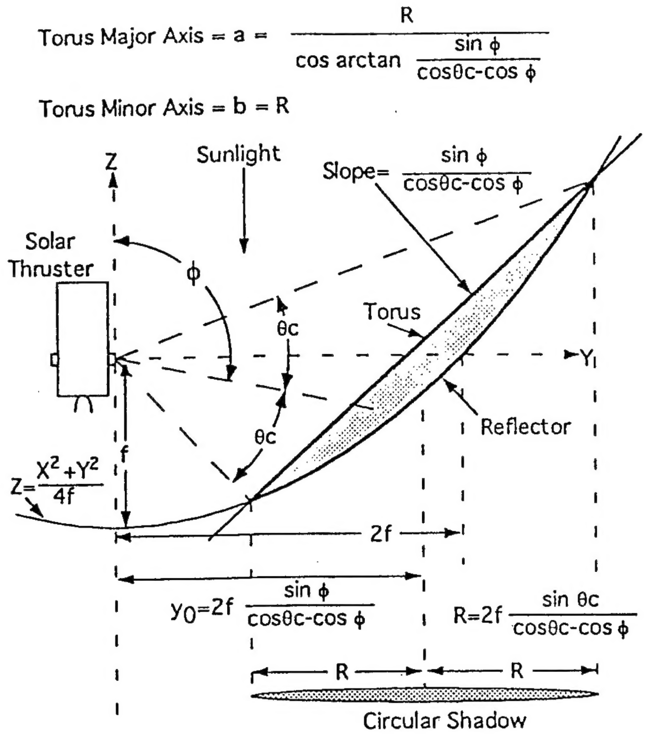


Figure 5: OFF-AXIS PARABOLOID GEOMETRY.

radius of this circle is R and given by;

$$R = 2f(\sin\theta_c/(\cos\theta_c - \cos\phi)) \tag{16}$$

The distance of the center of this circle from the Z axis is y_0;

$$y_0 = 2f(\sin\phi/(\cos\theta_c - \cos\phi)) \tag{17}$$

The torus has a minor axis length b equal to R. The major axis length a is given by;

$$a = \frac{R}{\cos\arctan(\sin\phi/(\cos\theta_c - \cos\phi))} \tag{18}$$

The area enclosed by the torus is πab and the perimeter p is approximately;

$$p = 2\pi\left(\frac{1}{2}a^2 + \frac{1}{2}b^2\right)^{1/2} \tag{19}$$

Given the desired power P collected by the concentrator, the desired concentration ratio R_c, and the tilt angle ϕ, then the off-axis concentrator geometry can be determined. The desired peak concentration ratio determines the cone half angle;

$$R_c = 46,000\sin^2\theta_c \tag{20}$$

The radius R is chosen to give the desired power P;

$$P = I_{earth}\pi R^2 \tag{21}$$

I_{earth} can be reduced to account for reflection losses. The choice of tilt angle ϕ is more complicated but not as critical. Tilt angles greater than 90 degrees give shorter major axes, less mass, and higher geometric concentration ratio; a tilt angle of 90 degrees centers the cone on the receiver better and possibly makes receiver design less complicated. For tight spacecraft fairings, tilt angles larger than 90 degrees may be required to minimize the length of the concentrator structure. Once ϕ is chosen, the focal length f is then calculated from Equation 16;

$$f = \frac{R}{2\sin\theta_c/(\cos\theta_c - \cos\phi)} \tag{22}$$

The length of the trusswork that connects the torus to the thruster can be calculated at this point. The three most popular mount points on the torus (but not necessarily the best) are the lower intersection of the major axis with the torus (see Figure 5) and the two intersections of the minor axis with the torus. These three points are usually connected to the thruster as close to the receiver aperture as possible onto a turntable mounting to allow the concentrators to rotate. The length of the two struts going to the minor axis is given by;

$$L_{minor} = \frac{y_0^2 + R^2 + 4f^2}{4f} \tag{23}$$

The strut going to the lower major axis intersection has its length given by;

$$L_{major} = \frac{(y_0 - R)^2 + 4f^2}{4f} \tag{24}$$

The perimeter and enclosed area of the torus were given earlier. For this simple system, given the linear mass density of the trusswork and the mass per unit area of the concentrator, the mass of a concentrator system can be estimated.

EXAMPLE: 2 X 25 KW RIGID CONCENTRATOR

For an sample case, consider the design of a spacecraft with a 50kW concentrator system consisting of two identical off-axis 25kW paraboloid concentrators. The value of the cone half angle is chosen to give a peak concentration ratio of one half the theoretical maximum. This condition is given by Equation 20 and is met when the cone half angle $\theta_c = 45°$. The shadow radius R is found from Equation 21, assuming a 10 percent reflection loss, to be 2.56 meters. A rather large

ıngle f=120 degrees is chosen to minimize the long of the torus since we are assuming a very tight fit ıe booster fairing. The focal length can then be de- ıined from Equation 22 to be 2.18 meters. Equation ives the offset of the center of the shadow from the of symmetry of the paraboloid; $y_0 = 3.14$ meters. minor axis of the torus is equal to R. The major a of the torus is $1.23R$ or 3.15 meters. The total th of the major axis is $2a = 2 \times 1.23R = 6.302$ ers.

he length of the major axis is almost short enough t in an Atlas fairing. If a rigid reflector is used, the ;traint of using a torus can possibly be relaxed and concentrator could be trimmed, then added to at tegic places to produce a better fit. The calculation the elliptical concentrator would simply serve as a le in this case to determine the approximate dimen- s and geometry for a rigid nontoroidal concentrator ɔort structure.

he mass m of the support structure is estimated by;

$$m = (2L_{minor} + L_{major} + p)\lambda_0 \left(\frac{R}{R_0}\right)^{\mu} \qquad 25$$

re λ_0 is a linear mass density characteristic of the t and torus material for a radius R_0 shadow, R is the us of the concentrator shadow, and μ is a real num- between 0.0 and 1.0. As the size of a concentrator em is increased, the linear density must also increase ɔrovide greater stiffness. $\mu = 1.0$ would give a lin- mass increase proportional to the concentrator area, ch should be avoided (e.g., structural stiffness can ncreased by making larger diameter, hollow struts ead). $\mu = 0.0$ represents no change in strut size with rger system which seems unlikely. The baseline case ɟ here will assume that a PL(OLAC) altitude cham- deployed foam inflated strut is stiff enough. This ıt is three inches in diameter and weighs approxi- ;ely 0.5 kg/m. Therefore, $\lambda_0 R^{\mu}$ is set to 0.5 kg/m. It lso assumed here that the torus and struts are made of the same material. The length $L_{minor} = 4.1$ me- ; by Equation 23. The length $L_{major} = 2.2$ meters Equation 24. The perimeter of the torus $p = 18$ me- ; from Equation 19. Therefore, the total mass of the port structure for one concentrator side is 28.5 kg Equation 25. The mass of the concentrator is ap- ximately the area enclosed by the torus times the :kness of the mirror times the mass density of the ror material (this assumes the added mass due to vature is negligible). For typical inflatable film con- trators, this mass works out to be one or two kg. As alternative, consider a rigid concentrator shell 1mm :k. The area of the concentrator is 25 square meters. If the reflector shell material density is 2g/ml, then the mass is about 50 kg.

RECENT CONCENTRATOR DEVELOPMENT EFFORTS

There have been several trade studies of concentrator structures performed. All of these studies reviewed considered either a 7 by 9 meter concentrator system, a 30 by 40 meter concentrator system, or both. This paper does not include previous studies since a review of these can be found in another paper (Laug, 1989). This set does reflect the most current information on structural issues regarding the accuracy of concentrators designed for solar-thermal propulsion. There is not yet much data on the optical performance of space-deployed concentrators.

The Gossamer Baggie Torus project, performed by Thiokol Corporation for the Phillips Laboratory(Lester and Warner, 1994), was a study of a concentrator support torus using FIR (Foam Inflated/Rigidized) structures. A proof-of-concept 8-foot diameter torus was deployed in an altitude chamber as a part of this project. Thiokol's favorite torus structure, as designed in this study, was a 7 by 9 meter elliptical truss structure.

The foam used to inflate the tubes of the FIR structure serves to expand the structure from a size that will fit into a booster fairing to the desired concentrator size. The tubes are made of E-glass cloth impregnated with ultraviolet curing resin. Ultraviolet radiation from the sun cures the tube walls after inflation. At the same time the inflation foam hardens. Stiffness is provided primarily by the rigidized tube walls. The foam increases the buckling strength by a factor of 2.3 in laboratory deployed tubes. The 8-foot diameter proof-of-concept torus began as a 2-foot diameter torus and completely deployed in about 15 seconds.

Producing an elliptical structure would require curved E-glass cloth tubes. While these are available, Thiokol considered the cost prohibitive. Instead, they choose to approach the elliptical shape with a dodecagon. This results in a mass only a few percent greater than an elliptical shape. Each side of the dodecagon consists of three parallel tubes instead of a single tube. This gives the required stiffness at minimum mass.

The total estimated mass for the Thiokol 7 by 9 meter design is 47 kg. This also includes the weight of the canisters which inject foam into the tubes during deployment. Thiokol did not calculate masses for the attachment struts from the torus to the spacecraft.

Thiokol and SRS Technologies are planning a rigidized-concentrator deployment in December 1994.

This deployment will use a FIR structure and an inflatable concentrator with two chambers. The first chamber is inflated with gas to form the appropriate shape for the concentrator reflective surface. The second chamber is foam-inflated against the pressurized back side of the reflective surface. Once the foam hardens, the front film of the front chamber can be removed and the concentrator shape is maintained by the hardened foam.

Harris (Borell, et al., 1993) has also studied concentrator support structures. They considered a 30 by 40 meter system, a 7 by 9 flight test version of the 30 by 40 meter system, and a Single Chamber Concentrator (Gierow, et al, 1992). Harris has been designing and building space deployable antennae and concentrators for the Air Force and NASA for many years. Some of their past ideas are considered in their study.

They based their torus deployment and support system on work done for NASA in the mid-1980s. A 15 meter diameter circular version of their torus (which they call a hoop) was actually fabricated for NASA-Langley. The struts that attach the concentrator to the spacecraft are graphite epoxy which telescope and latch upon deployment. The torus consists of 44 segments of 2.5 inch diameter graphite-epoxy tubes which unfold accordion-style by the action of small motors in each hinge. The number of segments was chosen to minimize stowage size.

The Harris 7 by 9 meter concentrator system will fit into a 28-inch diameter cylinder that is 28 inches high. The calculated mass of the total system is 57 kg. This mass includes everything but the turntable to rotate the concentrator about its axis. Under zero acceleration, this concentrator will generate a peak intensity of about 13,000 suns which is sufficient for most thruster designs. Under $1g$ of acceleration (considered for ground testing), the peak intensity remains greater than 12,000 suns. The single chamber 7 by 9 meter system weighs about 23 kg and will fit inside a GetAway Special (GAS) cannister on the Space Shuttle. However, this system is not as mature as the torus structures. No estimates were made of optical performance in this report.

The Harris 30 by 40 meter concentrator system is the same basic design as for the 7 by 9 meter system except that the torus will consist of 78 segments. This system will stow to 65 inch diameter by 65 inch length. The mass is estimated to be 280 kg.

Martin Marietta has proposed using graphite/gelatin as the torus material (Draper, 1993) (preferred over kevlar/gel) for a 30 by 40 meter concentrator support system. This type of torus is inflated with gas and then rigidized to form a cylindrical shell. They also considered a rigid, folding torus similar to that proposed by Harris. This particular report also considered in some detail mission requirements and pointing and tracking issues. In addition, space environmental hazards are addressed.

Martin Marietta compared 10 variations of the two basic approaches in the previous paragraph and selected a rigid torus with four telescoping, graphite/epoxy tube struts. This system meets a $\pm 0.5°$ pointing criteria under acceleration and will package into the Space Shuttle cargo bay. The mass of this system is estimated to be 430 kg. The weight can be reduced considerably using a graphite/gelatin inflatable self-rigidizing torus. This system would weigh about 260 kg. This system has a higher developmental risk and was therefore Martin Marietta's second choice.

TRW Space and Electronics Group selected inflation/rigidized struts and torus to support an inflatable concentrator (Kovalcik, 1993) . The 120-inch diameter Space Station Alpha Joint is suggested for a concentrator attachment point and to allow rotation about the axis perpendicular to the spacecraft. It has been prototyped already and appears to be suitable for a Solar-Propulsion spacecraft. There are three attachment points on the torus and three attachment points on the Alpha Joint. The Alpha Joint is connected to the concentrator by six struts arranged as three bipods. The two struts of a bipod meet only at the concentrator. The estimated total weight of one concentrator and support (minus the Alpha Joint) is 205 kg. The TRW configuration gives less than 2.2 milliradian slope error. To do this requires the strut materials have a low coefficient of thermal expansion, such as given by Kevlar.

Lockheed has studied the widest variety of strut types of all PL funded trade studies to date (Richie, 1993) . These types are IRSC (Inflated, Rigidized, Space-Cured Composites) tubular, Lenticular, Inflated Pressurized tubular, Bi-STEM (Storable, Tubular, Extendible Member), Coilable, FAST (Flexible Articulated Square Truss), and Telescopic. For each type of strut, mass and volume were estimated. Some of these need thermal protection to reduce thermal expansion effects. Five additional criteria were used in their selection of best strut; material shelf-life, ground test deploy simulation, deployment location accuracy, material dimensional stability in space, and damage tolerance in space. Lockheed assumes that the torus is inflated and then rigidized so this report does not discuss the torus in great detail.

Lockheed deemed four strut concepts worthy of further study; Lenticular, Coilable, FAST, and Telescopic, all of which used composite materials. They considered FAST to be the most accurate during deployment, to have low developmental risk, and to be one of the light-

strut concepts. The weight of all six FAST struts both sides of a 30 by 40 meter concentrator system 6 kg. The total weight of both concentrators with port structure, including turntables, is about 450 kg. nting deflections due to accelerations was held to less n 0.5 degrees in their design.

'he Lockheed numbers are based on the assumption t thrust can be throttled up slowly so as not to ite structural vibration modes with a sharp thrust ısient. Furthermore, Lockheed assumes that the uster is throttled so that the acceleration never exds 0.002 g. This in fact may be a reasonable conıint, since at LEO (Low Earth Orbit) the propellant ks will be full and the acceleration will be low at thrust. Once the orbit has been raised sufficiently, propellant tanks will be lighter and thrust will need be reduced. However, low thrust in higher orbits :s not lengthen the trip time too much because thrust ıes can be longer since orbit periods are longer. Fur-:rmore, lower thrust will in general allow increased :cific impulse.

Figure 6 shows a plot of mass vs. minor axis di-ıeter for the above concentrator system studies. It ;een that current estimates between different groups : within about fifty percent of each other. All of these ısses are a reasonably small fraction, between 10 and percent, of a typical payload. While somewhat larger mass than conventional thrusters, the increased Isp ows a much lower propellant mass, on the order of percent less. The net result is a nearly doubled payıd. The 50 percent variation in estimated masses for ncentrator systems will appear as a 5 to 10 percent riation in payload.

The two curves on Figure 6 are based on Equation with linear mass density $\lambda_0 = 1.2$ kg/m. The top rve has $\mu = 0.4$ and the bottom has $\mu = 0.1$. The lue of λ_0 corresponds to the 7 by 9 meter concentrar and both curves are fit to the one point on Figure 6 r this size concentrator. The value for λ_0 is 2 to 3 nes larger than any strut designs for this size concenator. However, this is because the masses on Figure 6 clude more than just the strut masses (e.g. foam inıtion cannisters, pivoting joints, etc.) fit to Equation ;. If it happens that these extra masses scale the same ay as the strut masses, then this equation will give a ;eful fit for concentrator mass estimation. As such, ıe value of μ for the two curves indicate that the con-:ntrator support strut linear density is approximately ·oportional to the fourth root of concentrator diamer. More data points are really needed however, since ıere is only one point at the 7 by 9 meter size.

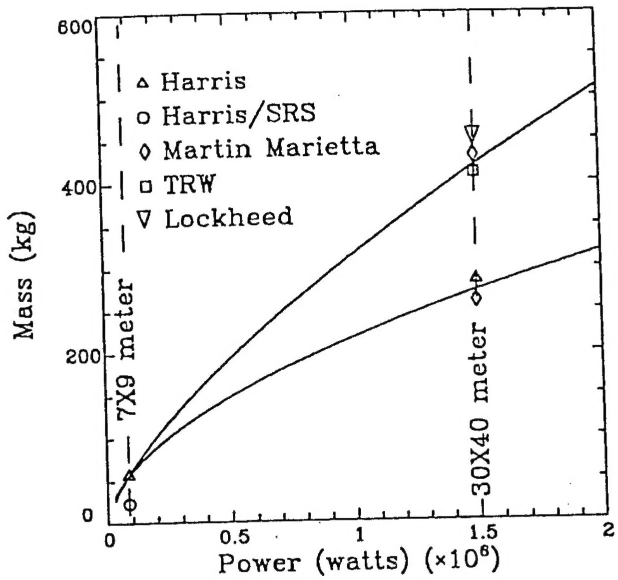


Figure 6: TRADE-STUDY MASSES OF SOLAR-THERMAL THRUSTER CONCENTRATOR SYSTEMS.

CONCLUSION

The maximum possible specific impulse Isp of a solar powered rocket is proportional to the eighth root of the peak concentration ratio. The thrust is proportional to the solar power collected and the energy efficiency of the absorber and concentrator. A peak concentration ratio of 10,000 to 1 should be sufficient to produce a thruster with an Isp of 9800 m/s(1000 s) and still retain reasonable thrust and efficiency. This is better than double that of conventional thrusters. To do this however, will require careful management of the asymmetric cone of focused light from an off-axis paraboloid concentrator.

Flight-weight concentrator developement is on-going as is the development of inflatable and inflatable/rigidized reflective films. Current data indicates that concentrators can be made light enough to realize the potential of Solar-Thermal Propulsion.

The Phillips Laboratory (OLAC) at Edwards Air Force Base is actively working to forward research and development in Solar-Thermal Propulsion. The concentrator support structure trade studies reviewed above were the result of PRDA Program Research and Development Announcement) contracts, for example. Recent funding is through SBIRs (Small Business Innovative Research), however, as money for larger contracts becomes rare. Topics such as space concentrator development, thruster development, pointing and accuracy,

etc., are of interest to the Phillips Laboratory Solar Propulsion Group.

The Phillips laboratory also has a 25 kW solar furnace on which to perform ground tests on thruster prototypes. This facility can also be used to test space based concentrator prototypes, provided they can be supported in a one gravity environment. Our goal is to achieve 9800 m/s (1000 seconds) *Isp* in the Phillips Laboratory Solar Lab.

References

Borell, G. J., Daubendiek, A. W., Updike, C. A., Dupper, J. K., Gierow, P., Truss Strut Trade Study, Final Report PL-TR-93-3008, 1993.

Draper, J. L., Gehling, R. N., Hanson, R. R., Shepard, D. F., Summers, J. L., Graphite/Kevlar^{TM} String Truss, Final Report PL-TR-93-3032, 1993.

Gierow, P. A., Clayton, W. R., Kromis, D., Concentrator Technology, PL-TR-92-3030, Sept. 1992.

Kovalcik, E. S., Solar Concentrator System Development, Final Report PL-TR-93-3025, 1993.

Laug, K., "The Solar Propulsion Concept is Alive and Well at the Astronautics Laboratory", CPIA Publication 515, Vol I, PP. 289-324, 1989.

Lester, D. M., Warner, M. J., Gossamer Baggie Torus, Final Report PL-TR-93-3040, 1994.

Merkle, C. L., Yu, Sheng-Tao, Solar Rocket Plume/Mirror Interactions, AFAL-TR-87-059, 1988.

Richie, G. L., Ambrose, J., Gavin-Fenton, L., Jensen G., Linke, R., Sharif, P., Tse, B., Solar Astromast Torus, Final Report PL-TR-93-3034, 1993.

Sutton, G. P., Rocket Propulsion Elements, 6th edition, John Wiley & Sons, Inc., 1992.

Nakamura, T. and Irvin, B. R., Optical Waveguide Solar Space Power Systems, *Proceedings of the 1992 ASME•JSES•KSES International Solar Energy Conference,* pp. 801-810, Hawaii, 1992.

Winston, Roland, "Nonimaging Optics", *Scientific American 264*, 1991)